CONTROLLED REENTRY

OF THE GAMMA RAY OBSERVATORY (GRO)

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1. ABSTRACT

Reentry of the GRO satellite must be controlled because it is expected that a great portion of the massive spacecraft would survive the reentry into the earth's atmosphere with the debris possibly causing harm to human life and property. The intent of this paper is to present a technique, results, and conclusion for a controlled reentry scenario for GRO. The planned impact would occur in an uninhabited portion of the South Pacific Ocean.

Two major areas were analyzed. First, targeting analysis examined conditions under which the orbital maneuver study was done. Finally, the debris scatter involved analyzing effects of the breakup of the spacecraft on the impact area. These two areas were the basis from which the controlled reentry study was accomplished.

2. BACKGROUND INFORMATION

The Gamma Ray Observatory (GRO) satellite is scheduled to be launched in the first quarter of 1990 by the Space Transportation System out of the Eastern Test Range. GRO is a relatively large spacecraft with weight of approximately 34,500 lbs. GRO's mission is to study cosmic gamma ray sources. This satellite is equipped with four detectors to obtain the gamma ray science. They are: the Oriented Scintillation Spectrometer Experiment (OSSE), the Imaging Compton Telescope (COMPTEL), the Energetic Gamma Ray Experiment Telescope (EGRET), and the Burst and Transient Source Experiment (BATSE).

GRO will be transported by the shuttle to an initial parking orbit. The onboard propulsion system will be used to raise it to its final mission orbit. Mission life time, that period where science data will be obtained, is required to extend at least 27 months. The first year will take the form of a full sky survey, viewing gamma ray sources for two week periods of time. The second year, not yet confirmed, may consist of concentrated viewing of a few targets. If sufficient orbit maintenance fuel remains the mission life may be extended. At the end of mission life, a controlled GRO reentry is required into a relatively unpopulated region of the earth, since it is expected that major portions of the structure will survive the intense heat and forces during its flight back to earth.

The relatively unpopulated region of the earth used in this study was taken from TRW GRO Mission Contract "Observatory Reentry Plan (Final)" (July 1985).[1] The area, noted as the nominal impact area, is outlined by the following islands:

Nominal Impact Area

Place	Latitude	Longitude
South Point of Hawaii	18.95 N	155.73 W
Christmas Island	1.87 N	157.33 W
Hiva Oa Island	9.75 S	139.00 W
Ducie Island	24.75 S	124.77 W
Easter Island	27.12 S	109.37 W
Isla Sala-y-Gomez	26.47 S	105.47 W
Isla San Felix	26.28 S	80.08 W
Lima Peru	12.05 S	77.05 W
Isla Isabela	0.63 S	91.45 W
Clipperton Island	10.28 N	109.22 W
Ocean Location	18.28 N	123.00 W

See Figure A-1 for pictorial representation of the impact region. For reference purposes the length for descending groundtracks measured from the southern point of Hawaii to Isla San Felix is approximately 9,500 kilometers, and the length measured form Hiva Oa Island to the coordinate (25 S latitude, 100 W longitude) is approximately 4,500 kilometers. This impact location is desirable because: it is the largest location within the +/- 28.5 latitudes that is comparatively uninhabited and consists mainly of open ocean; also, it is a location that is achievable without the additional fuel penalty of a plane change. The most favorable targeting will produce an impact that is centrally positioned within the impact region and therefore farthest from the islands tabulated above.

As a result of the location of the nominal impact box, Tracking Data Relay Satellite System (TDRSS) coverage was a concern. TDRSS is the environment with which contact with the GRO satellite will be maintained. The two TDRS's (East and West) positioned at 41 degrees West longitude and at 171 degrees West longitude respectively, resulting in each having an exclusion region in which contact can not be established between that particular TDRS and a user spacecraft. The inter-section between each TDRS's exclusion region is known as the TDRSS Zone of Exclusion (ZOE). The TDRSS ZOE for a spacecraft at an altitude of 300 km has East longitudes between approximately 58 degrees and approximately 95 degrees. The GRO maneuvers are initiated approximately between 55 to 100 degrees East longitude which contain the region where TDRSS coverage is not readily available. In addition, the lower the altitude of the spacecraft the larger the ZOE region gets. As a result, some maneuver burns will have to be initiated with stored commands and communication will have to resume when GRO is out of the ZOE region.

The following pages represent the analysis performed and the results obtained using the above information as a basis for the GRO controlled reentry.

3. TARGETING ANALYSIS

Reentry Targeting Analysis was done to investigate a feasible technique for a controlled GRO reentry into an unpopulated region of the earth. Three areas were be addressed under this analysis item. They are: assumptions, methodology, and TDRSS coverage. Each of the above items played a significant role in the targeting analysis and is discussed below.

3.1 Assumptions

The starting reentry orbit used was circular, approximately (315 km x 315 km). This is assumed to be the altitude where an STS rendezvous would take place and the remaining usable fuel on board is at least 1000 pounds to be used for the controlled reentry. The orbital conditions chosen for the controlled reentry were obtained from Code 554 GRO Lifetime Studies. This orbit represents a likely candidate for GRO at the end of life phase of the mission. To model the atmospheric conditions, solar flux obtained from the 97th percentile Marshall Flight Center Prediction Table August 1987, was used. Listed below are the assumed orbital elements and solar flux that were used for this study:

OSCULATING ELEMENTS

Epoch	April 1, 1992	11 hr. 19 min. 43 sec. GMT
semimajor axis	6695.389	km
eccentricity	0.000143	
inclination	28.51771	deg.
node	0.000013	deg.
argument of perigee	19.60527	deg.
mean anomaly	340.3947	deg.
solar flux	200.0	W/M sq. * Nt.-M

The major software tools that were used for this analysis were the General Maneuver (GMAN) program and the Goddard Mission Analysis System (GMAS) program. GMAN was used to compute the orbital maneuvers. GMAS was used for propagating from the post-burn state vector. For this analysis GMAS was equipped with an atmospheric density model that took into consideration the increased drag effect of near earth conditions.

In order to perform burn simulations, GRO-unique spacecraft parameters (especially the propulsion system) were modeled. The GMAN program, in modeling the GRO satellite at the beginning of controlled reentry, was given the following spacecraft parameters:

Spacecraft Parameters

Total Weight	31182.0 lbs
Total Expendable	1000.0 lbs
Fuel Weight	1040.0 lbs (260.0 lbs per tank)
Fuel Pressure	105 lbs per square inch absolute
Fuel Temperature	24.0 degrees Celsius
Cross Sectional Area	46.0 square meters
Drag Coefficient	2.2

The combined effects of these two mission analysis programs, GMAN and GMAS, provided good estimations of the orbital maneuvers and the orbital evolution.

The fuel considerations for each phase of the mission were obtained from TRW Gamma Ray Observatory Mission Contract based on a 34,500 lbs spacecraft at liftoff with 40 lbs of residual fuel.

C-

Phase	Fuel (lbs.)
Ascent	1315
Orbit Maintenance	445
Reentry	1000
Rendezvous	1040
Attitude Control	400
Total	4200

It will be assumed that the fuel load at the onset of reentry will be 1,040 lbs; 1000 lbs usable fuel and 40 lbs of residual. The predicted fuel to be expended for maneuvers was obtained by transforming the Rocket equation:

$$\text{Delta-V} = g * Isp * \ln (Mo / (Mo - \text{Delta-M}))$$

to $$\text{Delta M} = Mo * (1 - e^{k}), \quad k = -(\text{Delta-V}) / g * Isp$$

where Delta-V is the velocity change, g is the gravitational constant (9.8 m/sec sq.), Isp is the specific impulse of the propulsion system, Mo is the total weight of the spacecraft at the start of the burn, Delta-M is the fuel weight expended. The results were verified by GMAN after each maneuver.

3.2 Methodology

The next tasks were to determine: what will be the maneuvering guidelines leading to the desired impact location; and what would be the best staging deboost altitudes. The combination of these two items established the methodology which was taken.

Addressing the latter of the two above items, it was obvious that an infinite number of staging altitudes are possible. However, to provide the fundamental procedures needed to handle this task, three scenarios were analyzed. A single maneuver, a 2-burn maneuver sequence, and a classical two and half Hohmann transfer were examined. Lowering only perigee was the approach for both the single maneuver and the two maneuver sequences. The single maneuver and the 2-burn maneuver sequences were the plans selected because of the tremendous fuel saving when compared to circularizing the orbit when deboosting. If the classical two and one half burn Hohmann transfer for circular orbits were used, a negative delta-V would be applied at apogee, which causes perigee to be lowered. A second, similar negative delta-V would be applied at perigee which causes the lowering of apogee, and the circularization of the orbit. The final maneuver would cause the spacecraft to impact. In comparison to the standard two and one half burn Hohmann, the lowering only of perigee entails applying a single negative Delta-V equal to the first burn in the two burn Hohmann case. Comparing the two methods, the lowering of only perigee has these advantages over the standard two and one half burn Hohmann transfer: there is a tremendous fuel saving (factor of 2, except for the terminal burn) which could be allocated to extend the mission life and operationally, it is simpler in that only one burn operation must be carried out per targeted deboost altitude, thus less risk. Hence, the Hohmann transfer will not be utilized for the controlled reentry.

Derived Maneuver Constraints:

1. Perigee altitude must be selected such that the spacecraft will reenter less than 1/2 revolution after the terminal burn.
2. All maneuvers were to be made approximately 1/2 orbit away from the impact location region as determined from debris scatter study.
3. A minimum of 24 hours will be allotted between multiple burns for orbit calibration and generating command loads.
4. The longitude and latitude at 50 kilometers altitude will be considered the longitude and latitude at impact which is based on GMAS test cases that produced tenths of a degree difference at 1 kilometer as compared to the 50 kilometers; there is practically a vertical drop.

Also, all terminal burns were targeted for an altitude of 50 km because it is well within the critical altitude span where the spacecraft will not skip out. A spacecraft with a perigee altitude above 90 kilometers may not reenter within the 1/2 revolution after the completion of the maneuver because the accumulated drag force may not be great enough to deplete the kinetic energy within the 1/2 orbit constraint. Thus, the risk factor is increased because the depletion of the kinetic energy from the orbit can occur anywhere and the spacecraft could impact in an unfavorable location. At the completion of the burn the satellite will not have reached a 50 kilometer altitude, however the earth's atmosphere will provide the needed drag to cause the remaining loss of kinetic energy for the spacecraft to be pulled in.

A tool was developed to aid in selecting the appropriate delta-V that causes the spacecraft to impact. This was achieved by manual estimations followed by fine tuning with the GMAN and GMAS programs. Several initial altitudes were examined targeting for the following perigee altitudes: 75, 50, 25, and 1 kilometers. The range from impact to the midpoint of the burn was computed. Delta-V and range were graphed for each case. Also, groundtrack plots of the initial orbit were produced. The groundtrack plots were used to determine which descending passes fell in the desired impact box and to back out a time for the maneuver to begin. When the desired groundtracks were chosen, usually there were three candidates, the placement of the maneuver was derived by backing away approximately 180 degrees. The range vs. delta-V graph was used as an indicator of how much change in velocity was needed. The combination of delta-V vs. range graph and the groundtrack plots gives a fairly good estimation of where the burn should begin to produce an impact in the desired impact region.

3.3 TDRSS Coverage

Analysis was done for each maneuver to determine if a line of sight contact could be established and sufficiently maintained between GRO and TDRSS during the burn. It was assumed for this analysis that the average maneuver for the first burn of the 2-burn multiple case will be a maximum of 7 minutes long and approximately 15 minutes long for the second of the 2-burn case; 22 minutes will be the burn duration for the single burn case. Analysis has shown that it may not always be possible to maintain TDRSS communications throughout a maneuver while performing the maneuver at the most opportunistic time to achieve the desired impact region. Due to the positions of the spacecraft at the time of the scheduled maneuver it may be mandatory to maneuver, without TDRSS coverage at all. Earlier analysis showed that an altitude of 315 kilometers is too high to affect controlled reentry with 1000 pounds of fuel. Thus, it was necessary to await orbital decay to 276 km or less. The analysis performed was based on maneuvering from altitudes ranging from 276 kilometers to 215 kilometers.

4. DEBRIS SCATTER

The area over which GRO debris is expected to scatter upon impact was studied. Knowledge of the scatter is required to determine the safe targeting range to ensure that all pieces impact within the designated region. The altitude assumed where the spacecraft will begin to break up is 83 kilometers; this is based upon TRW's Gamma Ray Observatory Mission Contract Observatory Reentry Plan (Final) July 1985.[2] Two areas of concentration were addressed, effects of the ballistic coefficients (BC) and calibration errors. Each was thought to cause some effect on the scatter.

One way of estimating the scatter was by computing the impact points over a range of ballistic coefficients (BC):

$$\text{Ballistic Coefficient} = (Cd*A)/(2*m)$$

where m is the mass of a particular object, Cd is the dimensionless drag coefficient, and A is the average cross-sectional area perpendicular to the velocity vector. "The BC is the measure of the spacecraft to overcome air resistance."[3] To put this in perspective, a satellite with a large BC (large area to mass ratio) will impact earlier than one with a small BC.

In addition to the ballistic coefficient, the efficiency of the thrusters was thought to play a significant role in the area over which the debris is scattered. Targeting errors are the effects of thruster calibration errors on the terminating reentry orbit. This was analyzed by inducing a +/- 10 percent calibration error using the GMAN targeting tool. The net result, as compared to the nominal, showed displacements. The +10 percent calibration error (firing hot) produced an impact further up track from the nominal; and the -10 percent (firing cold) produced an impact further down track from the nominal. Firing hot caused a 10% increase in the fuel consumed and the converse was true for the cold firing. Combining the thruster inefficiencies with the ballistic coefficients factors (hot with maximum and cold with minimum) also increased the scatter range. As expected, the hot firing and maximum BC impacted furthest up track in comparison to all of the previous cases, and conversely for the cold and minimum BC. Therefore, the determination of the maximum range over which the debris is expected to be scattered was based upon a range of ballistic coefficients in conjunction with thruster inefficiencies.

The BC range consisting of a maximum (135 x 10 -1 lbs/ft. **2) and a minimum (0.5 x 10 -1 lbs/ft. **2) corresponds to the high gain antenna and EGRET respectively. These coefficients were derived from TRW's study on potential spacecraft breakage. They are representative bounds for determining the range over which the debris is expected to scatter. The minimum BC causes a later impact and the maximum BC causes an earlier impact. Results obtained using the minimum BC and the maximum BC determined the lower and upper bounds over which the GRO will be scattered upon impact. The actual length of this scatter along the groundtrack is the arc distance along the surface of the earth formed by the vectors that extend from the center of the earth to the spacecraft impact points corresponding to the cold/minimum and hot/maximum BCs. The arc-length distance was computed using the following equations:

$$\cos \;= \sin\delta_1 \sin\delta_2 + \cos\delta_1 \cos\delta_2 \cos(\alpha_1 - \alpha_2), \quad 0 < \theta_i < 180 \qquad \text{(A-1)}$$ [4]

where (α_i, α_i) are the longitude and latitude points on a unit sphere and θ_i is the angle between them

$$S = \theta * Re \qquad \text{(A-2)}$$

where S is the arc-length distance of angle θ and Re is the earth's radius. This is consideted to be the along track scatter. Previous studies have shown that the cross track scatter is considered to be negligible.

5. <u>STUDY</u> <u>CASES</u>

5.1 Single Burn Scenario

A single burn scenario consists of one long burn that is applied to the spacecraft to lower perigee far enough so that the accumulated drag forces would deplete its kinetic energy sufficiently to cause it to impact. Analysis was performed for several single deboost maneuvers; however, only one will be presented. It is for an approximate (250 km x 250 km) circular orbit targeting to a perigee altitude of approximately 50 kilometers, this assumes a nominal spacecraft area, mass, and thruster efficiency. This orbit was achieved by allowing the spacecraft to drag down from the previously stated 315 km circular orbit for approximately 2 months. The reason for the desirable decay of the orbit was that the allotted reentry fuel of 1000 pound could not accommodate a controlled reentry to impact. There simply is not enough fuel to cause a designated impact from that altitude. Therefore, a lower altitude was required. The Delta-V needed to go from 250 km to 50 km was estimated by using the Vis-Via equation for the initial and final orbits and taking the difference between the two orbit velocities to find the impulsive delta between the two. The equations are as follows:

$$V1 \sqrt{Mu\ (2/R1 - 1/A1)}, \qquad V2 \sqrt{Mu\ (2/R1 - 1/A2)},$$

$$\text{Delta-V} = V2 - V1$$

where V1 and V2 are the velocities of the initial orbit and final orbit respectively, Mu is the gravitational constant, A1 and A2 are the semimajor axes of the initial and final orbits, R1 is the radius of apogee of both initial and final orbits. Once the Delta-V was calculated it was applied at a specific ignition time and direction for this single deboost maneuver which would take place approximately 1/2 orbit away from the desired impact point. In addition to the 1/2 orbit requirement for burn placement, the Delta-V was applied near an ascending node such that the spacecraft reentered near a descending node. Reentry near a descending node is important in order to achieve a groundtrack pass along the length of the impact area region, as opposed to reentering near an ascending node along the shorter width of the impact region.

Table A-1 lists the effects of the nominal, +/- 10 percent calibration, and the maximum and minimum BC cases in terms of ignition and impact coordinate points for opportunities 1, 2, and 3.

Table B-1 is a list of the debris scatter.

Figure A-2 show the groundtracks for the three consecutive opportunities. Impacts are recorded for firing 10% hot and cold, with a maximum and minimum BC, and the combination of the cold/minimum and hot/maximum are displayed.

In essence it appears that the size and weight of the spacecraft and the efficiencies of the thrusters are important factors to look at in determining the area over which the debris is scattered. The total length of the debris scatter measured from the hot/max impact to the cold/min impact is approximately 3,100 kilometers long. The fuel consumption for all of the cases fell marginally within the fuel allotment of 1000 pounds. Orbital conditions achieved were suitable to drive the spacecraft well below the critical skip altitude. Finally, the lengths of the burns were approximately twenty-two minutes initiating very close to the ZOE region. Thus controlled reentry is feasible with one long single burn.

5.2 Two-Burn Scenario

The 2-burn reentry scenario entails progressively reducing the perigee altitude by dividing the maneuver over two separate orbits over 24 hours until perigee is lowered well below the critical skip-out attitude. This two burn strategy was considered since the burn error produced by sequential retargeting should be smaller. Any number of burns could be made, however the goal is to obtain a degree of accuracy but yet be efficient. The 2-burn maneuver sequence does this. The 2-burn maneuver sequence, as the single burn maneuver sequence begins at an altitude lower than 315 km. The 2-burn maneuver sequence was started at a near circular (288 km x 276 km) orbit. It took a little over one month to decay down from 315 km. Establishing where to begin the burns for each of the two deboost orbits was based upon the alignment of the line of apsides (the diameter from the apogee point to the perigee point) with the impact box. The placement of the burn was also near an ascending node. The natural precession of the orbit was taken into account for placement of the first burn so that reentry could occur approximately 24 hours later. The orbit precessed approximately 7 degrees per day. Therefore, the first burn was performed approximately 7 degree away from the desired location for apogee. One of the maneuvering guidelines states there will be at least 24 hours between burns. The first burn was targeted for a perigee height of approximately 215 km. The Flight Dynamics Division has performed studies showing that below 200 km altitude there is a possibility that nominal attitude control of the spacecraft may be lost. It is imperative to maintain good attitude control before, during and after the maneuver especially since another maneuver is needed to drive in the spacecraft. A perigee altitude of 215 km has a padding to allow for a one day decay of the orbit and other complications that may occur. From 215 km there are approximately seven days before a portion of the orbit slips below 200 km. The second burn, which causes the spacecraft to impact was performed one day after the first burn. Also, the second burn is performed at apogee, near an ascending node. As stated earlier there are three consecutive orbital opportunities for impact into the nominal impact region and the same logistics apply. This second burn was targeted for a perigee altitude of approximately 50 kilometers. Like in the single maneuver scenario, analysis was performed using +/- 10% calibrations. However, after performing the +10% calibration case the results indicated that a lower perigee altitude was achieved on the first burn and more fuel was expended. The second burn could not achieve the total burn time desired because the fuel ran out. Therefore, all of the cases were tempered by the delta burn time (minutes) that was needed for the hot case to achieve a low enough perigee altitude which would result in an impact within 1/2 revolution and not run out of fuel before the completion of the maneuver. This delta burn time affected the nominal targeted perigee altitude of 50 kilometers. The perigee altitude for the nominal case was approximately 20 kilometers higher and the cold was even higher. However, all of the cases resulted in the designated area impacts within 1/2 of a revolution.

Table A-2.1 and A-2.2 list the effects of the nominal, +/- 10 percent calibration, and the maximum and minimum BC cases in terms of ignition and impact coordinate points for opportunities 1, 2, and 3.

Table B-2 is a list of the debris scatter.

Figure A-3 shows the terminating groundtracks for the three consecutive opportunities. Impacts are recorded for firing 10% hot and cold, with a maximum and minimum BC, and the combination of the cold/minimum and hot/maximum.

Figuure A-4 is an altitude vs time graph showing apogee and perigee decay against time.

5.3 Evaluation of cases

The two cases represent two slightly different methods. The 2-burn case allows for the calibrating of the thrusters and setting up the spacecraft for the optimal orbital conditions required for accurate targeting. Also, during the actual mission the errors due to targeting at impact will be somewhat smaller because the second burns can take into account the error caused by the first burn. The objective for this study was to see if a worst case scenario would provide for a controlled reentry where all the debris would fall in the box; the results did show this. However, there are uncertainties associated with any burn case. The thrusters could fail during a maneuver, unanticipated torque on the spacecraft could throw off the attitude control and misalign the direction of the thrust, or there could be any number of unexpected phenomena. However, after examining the results of these two cases the two burn appears to be better. The two burn case allows for the error in the first burn to be removed during targeting for the second burn, thus cutting the error down significantly from the single burn scenario. The single burn scenario does not allow for the calibration of the thrusters. Therefore, the 2-burn maneuver sequence is recommended as the most effective way of performing the controlled reentry of GRO.

6. CONCLUSION

The controlled reentry analysis of the 34,500 pound, Gamma Ray Observatory is a dynamic task filled with many subtle uncertainties and technical lessons. The paper represents a basis from which more detailed analysis will be done. In performing this analysis several other areas to investigate surfaced; e.g using another impact region located in the Indian Ocean, what affect will lift have on the spacecraft's deboost, as well as the flight path angle. As a result, this study has served as a catalyst by stimulating questions which will help further complete development of a controlled reentry program for the GRO spacecraft. Several of the assumptions made at the onset of the study have changed, and undoubtedly, some will even change as late as two years into the mission. However, the lowering of perigee technique presented here is a viable one (the 2-burn maneuver sequence is the recommended scenario). It is based on normal orbital occurrences; therefore, it is believed that the uncertainties about the orbit, spacecraft, and atmospheric conditions should not affect the foundation on which the analysis is based in providing a controlled GRO reentry.

If the reentry phase begins with GRO's altitude greater than 276 km, it is necessary to allow the orbit to decay to less than or equal to 276 km to accomplish the controlled reentry with less than or equal to 1000 pounds of fuel during a 2-burn scenario. Also, the reentry area is approximately 180 degrees away from the TDRSS ZOE. It will be necessary to sacrifice TDRSS coverage during the maneuvers to accomplish the controlled reentry into the designated region.

9. FOOTNOTES

1. TRW, "Observatory Reentry Plan (Final)", (July, 1985), 15.

2. ibd.

3. Wertz, James R., "Spacecraft Attitude Determination & Control", (Boston: D. Reidel Publishing Co., 1980), 64 .

4. ibd.

10. REFERENCES

1. Bate, Mueller, White. "Fundamental of Astrodynamics", NY: Dover Publications, Inc., 1971.

2. Jensen, Townsend, Kork, and Kraft. "Design Guide to Orbital Flight", New York: McGraw-Hill Book Co., 1980.

3. Wertz, James R. "Spacecraft Attitude Determination & Control", Boston: D. Reidel Publishing Co., 1980.

4. Computer Science Corporation/Contract NAS5-27888/64600, "Gamma Ray Observatory (GRO) Compendium Of Flight Dynamics Analysis Reports", R. McIntoch, December 1986.

5. TRW, "Mass Properties Status Report Gamma Ray Observatory", June, 1985.

6. TRW, "Observatory Reentry Plan (Final)", July, 1985.

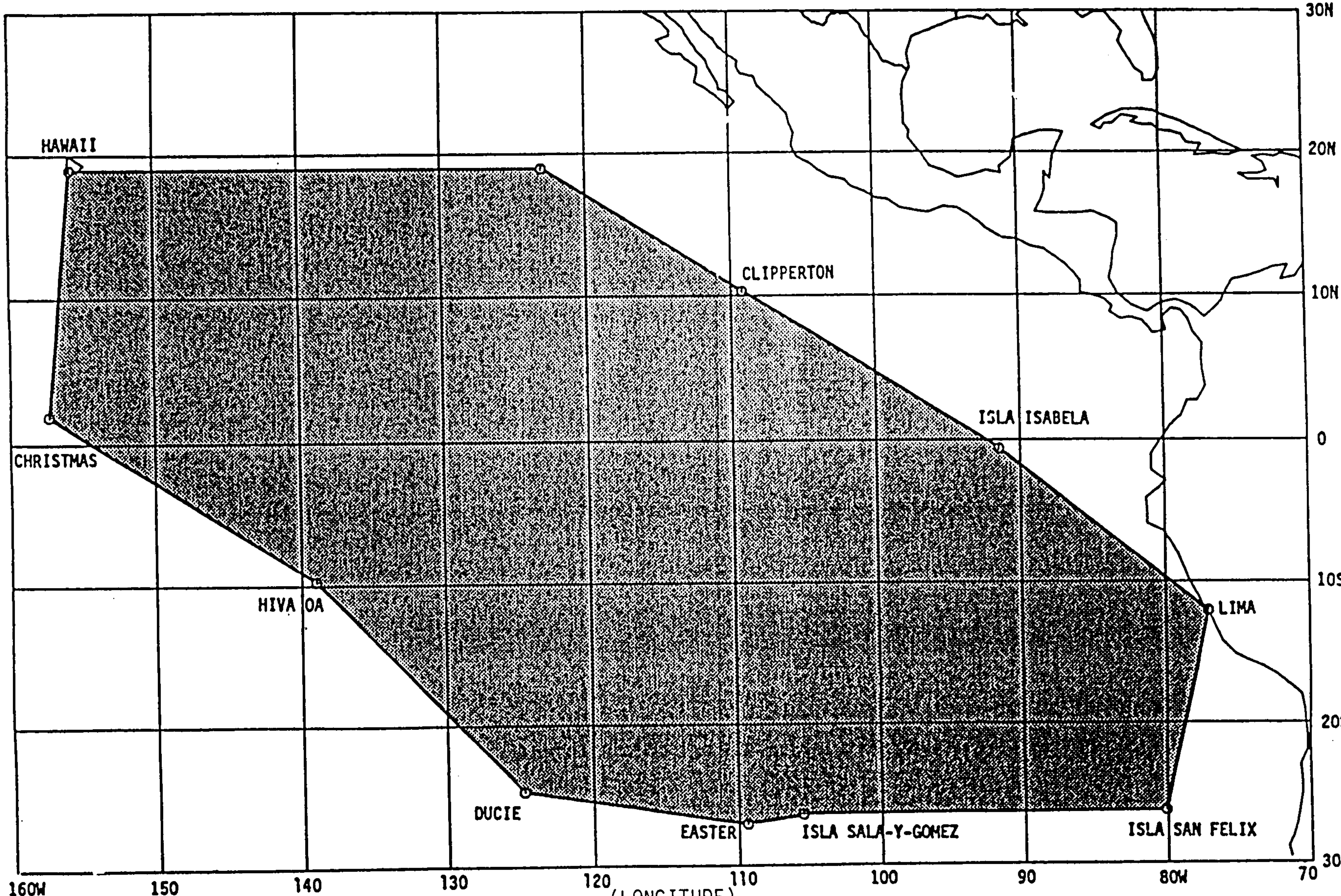

Figure A-1 Nominal Impact Area

Table A-1. Single Maneuver Sequence

POST MANEUVER CONDITIONS

OPPORTUNITY		COLD/ MIN BC	MIN BC	COLD	NOMINAL	HOT	MAX BC	HOT/ MAX BC
1	FUEL(LBS)	811	895	811	895	977	895	977
	BURN TIME (MIN)	22	22	22	22	22	22	22
	DELTA-V (FT/SEC)	193	213	193	213	233	213	233
	APOGEE(KM)	229	228	229	228	227	228	227
	PERIGEE(KM)	64	46	64	46	26	46	26
2	FUEL(LBS)	811	895	811	895	977	895	977
	BURN TIME (MIN)	22	22	22	22	22	22	22
	DELTA-V (FT/SEC)	193	213	195	213	233	213	233
	APOGEE(KM)	229	228	229	228	227	228	227
	PERIGEE(KM)	64	45	64	45	26	45	26
3	FUEL(LBS)	811	895	811	895	977	895	977
	BURN TIME (MIN)	22	22	22	22	22	22	22
	DELTA-V (FT/SEC)	193	213	193	213	233	213	233
	APOGEE(KM)	229	228	229	228	227	228	227
	PERIGEE(KM)	66	47	66	47	28	47	28

Table B-1.

SINGLE MANEUVER SEQUENCE	DEBRIS SCATTER (KM)
NOMINAL TO HOT/MAXIMUM BC	1700
NOMINAL TO MAXIMUM BC	900
NOMINAL TO HOT	1100
NOMINAL TO COLD	1300
NOMINAL TO MINIMUM BC	150
NOMINAL TO COLD/MINIMUM BC	1400
*HOT/MAXIMUM TO COLD/MINIMUM BC	3100

DEBRIS SCATTER

GMT = 1986 :4: 3:12:34:56.789

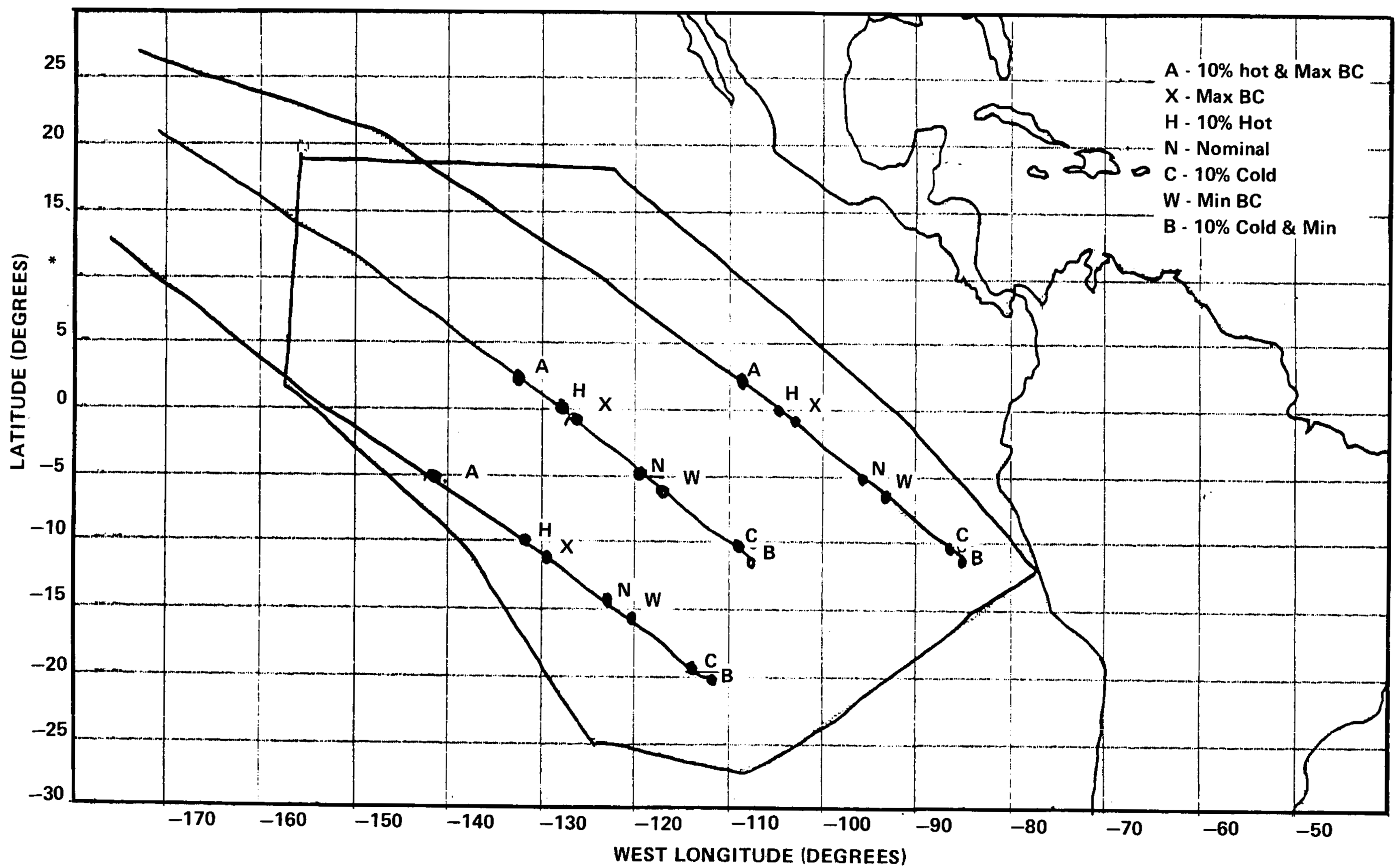

Figure A-2. Single Burn Maneuver

Table A-2.1 2-Burn Maneuver Sequence

FIRST OF TWO POST MANEUVER CONDITIONS

OPPORTUNITY		COLD/ MIN BC	MIN BC	COLD	NOMINAL	HOT	MAX BC	HOT/ MAX BC
1	FUEL(LBS)	270	298	270	298	326	298	328
	BURN TIME (MIN)	7	7	7	7	7	7	7
	DELTA-V (FT/SEC)	64	70	64	70	77	70	77
	APOGEE(KM)	276	276	276	276	276	276	276
	PERIGEE(KM)	222	215	222	215	208	215	208
2	FUEL(LBS)	270	298	270	298	327	298	327
	BURN TIME (MIN)	7	7	7	7	7	7	7
	DELTA-V (FT/SEC)	64	70	64	70	77	70	77
	APOGEE(KM)	276	276	276	276	276	276	276
	PERIGEE(KM)	222	215	222	215	208	215	208
3	FUEL(LBS)	270	298	270	298	327	298	327
	BURN TIME (MIN)	7	7	7	7	7	7	7
	DELTA-V (FT/SEC)	64	70	64	70	77	70	77
	APOGEE(KM)	276	276	276	276	276	276	276
	PERIGEE(KM)	222	215	222	215	208	215	208

Table A-2.2 2-Burn Maneuver Sequence

SECOND OF TWO POST MANEUVERS CONDITIONS

OPPORTUNITY		COLD/ MIN BC	MIN BC	COLD	NOMINAL	HOT	MAX BC	HOT/ MAX BC
1	FUEL(LBS)	528	578	528	578	630	578	630
	BURN TIME (MIN)	14.7	14.7	14.7	14.7	14.7	14.7	14.7
	DELTA-V (FT/SEC)	126	138	126	138	151	138	151
	APOGEE(KM)	264	261	264	261	256	261	256
	PERIGEE(KM)	89	72	89	72	54	72	54
2	FUEL(LBS)	528	578	528	578	630	578	630
	BURN TIME (MIN)	14.7	14.7	14.7	14.7	14.7	14.7	14.7
	DELTA-V (FT/SEC)	126	138	126	138	151	138	151
	APOGEE(KM)	263	260	263	260	256	260	256
	PERIGEE(KM)	89	72	89	72	54	72	54
3	FUEL(LBS)	528	578	528	578	630	578	630
	BURN TIME (MIN)	14.7	14.7	14.7	14.7	14.7	14.7	14.7
	DELTA-V (FT/SEC)	126	138	126	138	151	138	151
	APOGEE(KM)	263	260	263	260	256	260	256
	PERIGEE(KM)	89	71	89	71	53	71	53

Table B-2.

2-BURN MANEUVER SEQUENCE	DEBRIS SCATTER (KM)
NOMINAL TO HOT/MAXIMUM BC	2300
NOMINAL TO MAXIMUM BC	800
NOMINAL TO HOT	1500
NOMINAL TO COLD	2100
NOMINAL TO MINIMUM BC	150
NOMINAL TO COLD/MINIMUM BC	2300
*HOT/MAXIMUM TO COLD/MINIMUM BC	4600

DEBRIS SCATTER

GMT = 1986 :4: 3:12:34:56.789

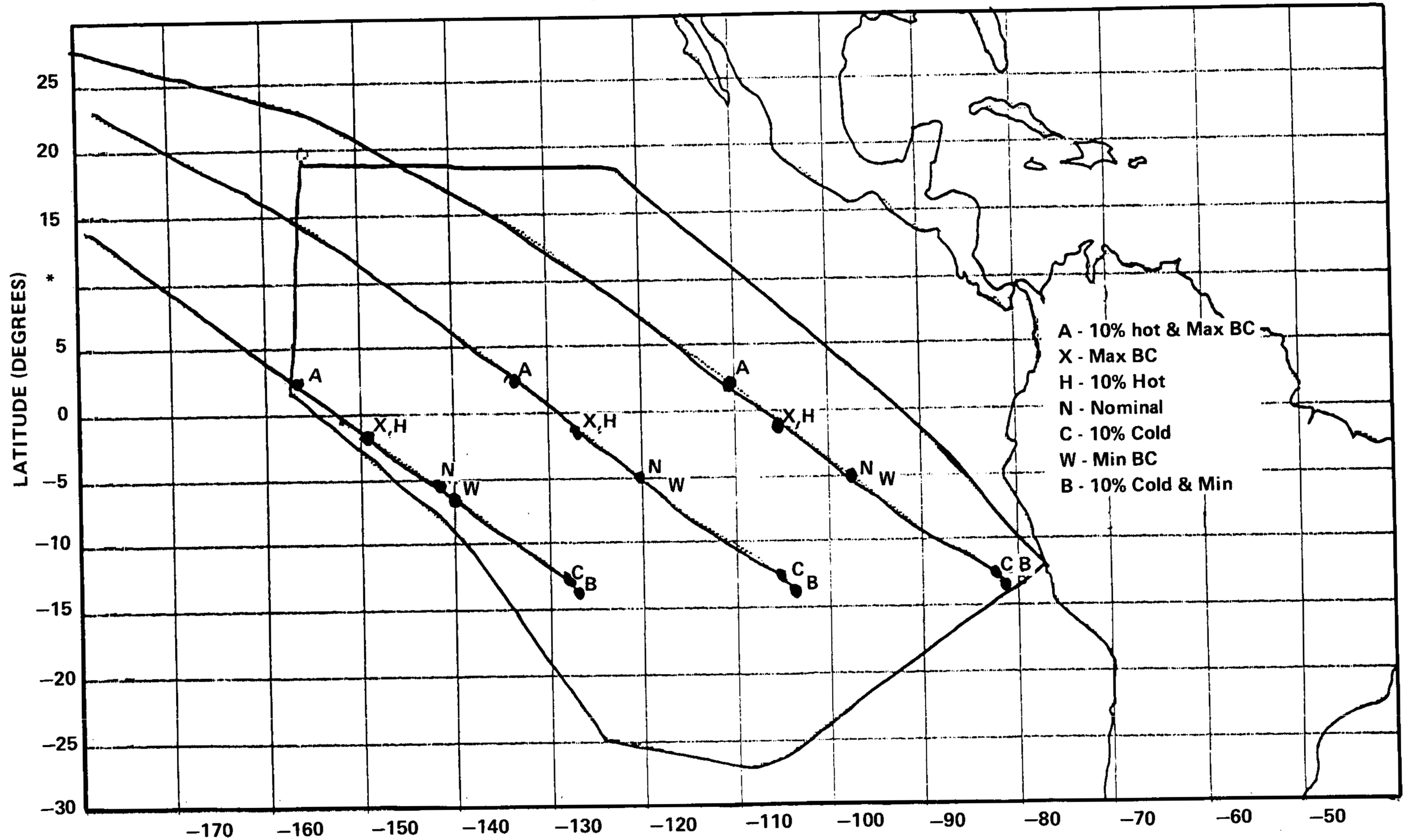

Figure A-3. 2-Burn Maneuver Sequence

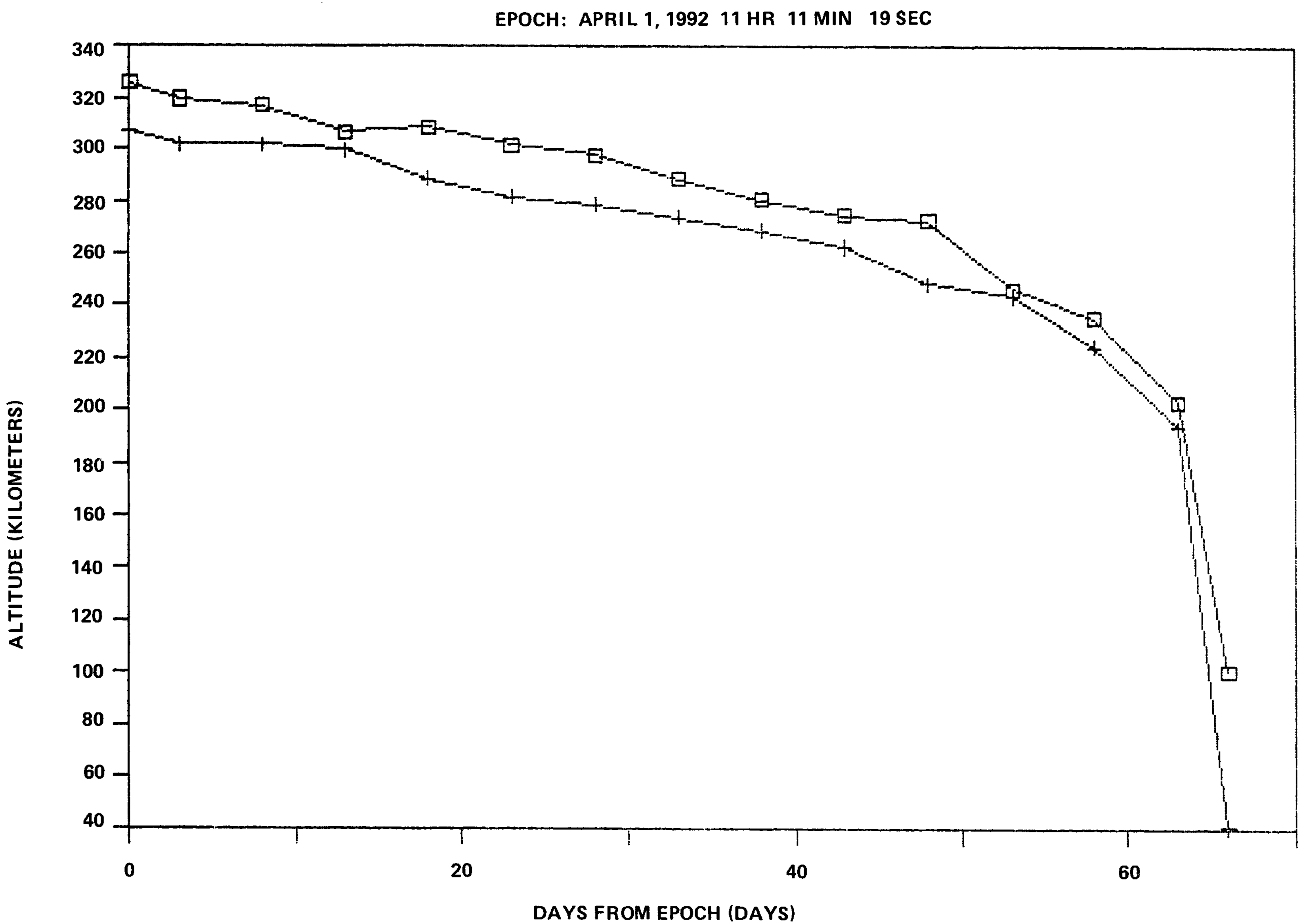

Figure A-4. Altitude vs Time Decay (315 km cir.)